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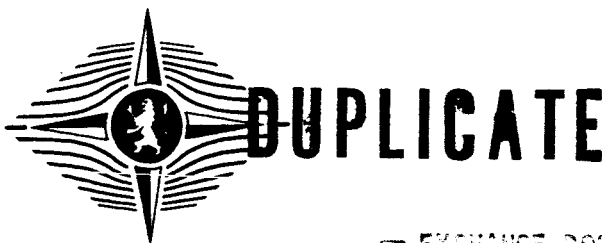
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THE ROLE OF AERONAUTICAL RESEARCH IN AIRPLANE DESIGN.

(A series of lectures given at Rhode Saint Genese February 1958)

By Axel T. Mattson.

of the National Advisory Committee for Aeronautics
of the U.S.A.



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THE ROLE OF AERONAUTICAL RESEARCH IN AIRPLANE DESIGN

INTRODUCTION

Aeronautical research accomplished many years ago supplied us with knowledge, techniques, and apparatus now considered the accomplishments of our modern age. For example, in Vienna in 1889, fourteen years before the Wright brothers' first flight, an Austrian professor used a schlieren optical system to photograph supersonic flow for the first time. A few years later his son employed the same system to take the first schlieren photographs of the flow of air through a strange new contraption called the wind tunnel. The two gentlemen to whom I refer, of course, were Dr. Ernst Mach and his son, Ludwig, both professors of physics.

While the contributions of these two men were neither more nor less in importance than other great scientific pioneers in aeronautics, their work is significant in view of the state of the art at the present time. Their contributions and connection with the present state of the art is symbolized by the fact that now their name is one of the most used words in aeronautical science. Their name and the conditions it now stands for - the modern refinements of their basic tools and their techniques - are now being used the like of which was never predicted.

Today we are sometimes misled by the erroneous conception of a world in which all technical frontiers are boundless. Because of the spectacular progress being made in aeronautics, it is easy to jump to the conclusion that anything can be accomplished by an expenditure of enough money and manpower. Although this idea when applied to science has some validity, a more sober view will show that realism requires that we recognize some of the boundaries and limits. In aeronautical

research, it has been important to delineate and understand these limits. It is only when we understand these limits that we are able to circumvent them and proceed without a large waste in money and manpower. It is of utmost importance that we understand the techniques of measurements in all fields.

Basic research and development research is important, but the use of our skills and tools for directed research is of greater importance, since it is always related to the actual hardware to be built. The spirit of cooperation, of working together, exchanging information, in a scientific environment is indeed a first prerequisite.

In this paper I will attempt to summarize and discuss research philosophy that has been used in the United States in effecting and directing the development of airplane designs and summarize in a general way, research development for such things as performance, propulsion systems, stability and control, flutter and buffeting. It is hoped that this talk will indicate in a general way the philosophy used in formulating the role of aeronautical research in airplane design.

RESEARCH PROBLEMS

Research problems, of course, involve the variations of many inter-related parameters. Therefore, research problems do not lend themselves to straightforward solutions. They have to be broken down into smaller problems and relatively simple solutions obtained. These solutions then are fitted together in a manner to provide a better understanding of the overall problem.

When airplanes were designed and flew at subcritical speeds, the wind-tunnel programs were devoted mainly to drag clean-up problems and stability and control. The problems arising from increased speeds were usually those of propeller slip stream interference and dynamic pressure effects. Of major concern was the (power on) effect as the greater speeds demanded more powerful engines. Power on effects became so severe that airplanes became unstable. Hinge moments became excessive and aerodynamic balance was needed. Structure being flexible, led to aeroelastic twist resulting in many problems too specific to mention. Nevertheless, as quickly as these problems were solved airplane speeds increased until during World War II they were encountering the phenomena found only in research laboratories - compressibility effects. It was here that aeronautical problems increased by order of magnitudes and again real frontiers had to be exploited. Compressibility effects resulted in large increases in drag and gave serious trouble with regard to stability and control. Large increases in stability, trim changes, loss in lift, control ineffectiveness, shaking, increase in control forces. All of these effects led to the well publicized "sonic barrier". As a result, changes in the airplanes features took place. Wings became thin, sweptback, and were of low aspect ratio. They were powered by the newly developed turbojet and the troublesome propeller gave way to the turbojet.

The "sonic barrier" was conquered by the use of twin wings, sweepback, and low aspect ratio aerodynamic refinements, and we began to uncover the even wider variety of problems covering a speed range which includes two different types of flow phenomena.

Subsonic and Transonic

We knew that transonic flow was a mixture of subsonic, supersonic shock-induced type, and theoretical treatment was difficult and experiment was in need of invention. The airplane was unpredictable and stability problems associated with separated vortex flow and high angles of attack were many. The fuselage became important because of high angle of attack and size compared to the wing. The installation of the turbo-jet engine resulted in a different mass distribution being confined to the fuselage. This introduced moments of inertia in pitch and yaw along the fuselage which were previously concentrated in the wing or roll. The density of the airplane was increased by virtue of greater wing loadings and operational latitudes resulting in problems associated with dynamic stability such as spinning and other problems. The stability and maneuverability during landing and take-off deteriorated and problems at low speeds were encountered.

Throughout a major portion of aeronautical history the subject of airplane handling qualities remained undefined. It was not until 1940 that this problem received enough attention to warrant any research problems. At this time, however, programs were instigated both in the United States and Great Britain designed to define satisfactory handling criterion for aircraft. As a result, flight data were collected by H.A. Soule in the form of pilots' opinion under many flight conditions together with measurements of airplane response, forces

and control positions. (TR 700) (Ref. 1).

In the United States, this program was continued and R.R. Gilruth of the National Advisory Committee for Aeronautics report in TR 755 (ref. 2) the results of a further survey. This Technical Report has since served as a basis for handling qualities specifications that were later adopted by the military services and AGARD. This work is continually being expanded and modified as more results on more types of aircraft are obtained.

These requirements are for the most part the same in other countries and, in general, are treated the same. For example, the requirements of dynamic motions such as the phugoid and short-period longitudinal mode, rolling, spiral, and short-period lateral modes are considered universally the same. Damping rate requirements given in time to damp to a given fraction of the initial amplitude are for the most part universally the same. Requirements for static-longitudinal stability associated with control force and position with speed were specified. Static lateral stability was associated with control position and ability to trim an asymmetric condition. Specific rate of roll was covered. Included are requirements for stall warning and spin recovery. Requirements for transonic stability and control problems are still in the development stage. Therefore, the establishment of these requirements has had and will have an important influence on our ability to delineate problem areas and plan research programs. It has led researchers to establish the relationship of the requirements to basic concepts and has in no small measure guided our wind tunnel, rocket, flight, and analytical research problems (Figs. 1, 2, 3, and 4).

PLANNING AND EXECUTION OF RESEARCH PROBLEMS

The planning of research programs is largely dependent upon previous experiences and knowledge dictating problems that need further study. Basically it is an art that exploits previous experience to guide the experimental and theoretical investigations. Research programs can, generally speaking, be divided into two rather broad categories. The first is that dealing with the understanding of fundamental phenomena. The second deals with the application of these fundamentals to a complete article. The first is considered to be of relatively greater importance.

Fundamental phenomena - For wings potential-flow solutions generally evolve from basic understanding of the flow field. For example, wings can be represented by a sheet of vortices, doublets, or sources with the stream line on the surface. The aerodynamic derivatives then can be obtained by summation of the normal forces, suction forces and possibly the tangential viscous force. However, experiments are needed to indicate the departure of the real flow and thereby define limits on the applicability of theory. The boundary-layer violates most of the basic assumptions when it separates. For practical cases, it seems always to separate. For thin wings, experiments have shown that laminar separation occurs near the leading edge because of the inability to maintain the very high suction pressure required for attached flow. Separation of this type is known as a separation "bubble". (Fig. 5). At subcritical Mach numbers, this separation "bubble" can occur at relatively low angles of attack. (Approx. 3° for $t/c = 0.04$). The chordwise growth of the bubble is a function of angle of attack and generally the flow circumvents the "bubble" and re-attaches with a turbulent boundary layer. As a result, prior to the stall, we find experimentally a slight increase in lift

curve slope. Therefore, the predicted potential leading-edge suction is considerably more than the real case.

For a two-dimensional wing when the Mach number is increased beyond the critical, the leading-edge flow becomes supersonic. Now, because of supersonic expansion, the flow can attach itself. The expansion terminates with shock which, if the speed is increased, will move rearward. Leading-edge suction disappears and the forces and moments are dependent upon the Mach number. So much for the simple case - that of the supercritical section characteristics. The flow about three-dimensional surfaces although dependent upon section characteristics to some degree are now subject to such geometric parameters as sweep, high taper, low aspect ratio that can dominate the flow pattern. Vorticity occurs along the leading edge of sweep wings, tip effects depend on taper, and tip distortions will, of course, influence a large portion of a low aspect ratio wing. The vortex pattern about finite wings is illustrated in figure 6. The vortex lines separate from the wing leading edge near the apex and follow the leading edge sweep turning downstream at the tip increasing in strength. Vortex separation presents a serious departure, of course, from considerations used in potential flow. And, therefore, we know that the use of calculations based on potential flow are limited to practically zero angle of attack.

And, it was the understanding of this phenomena for experimental evidence which led Michael and Brown to treat the separated case and obtain a potential flow solution for triangular wings on the basis of slender body theory. The results predicted an increase in lift and experiments have given a qualitative verification. Vortex separation strongly influences stability derivatives and can also influence dynamic characteristics by time lag for establishment of hysteresis effects.

Long slender bodies inclined to the airstream lead to minimum pressure points above the longitudinal axis for the forward part and below the axis for the rear part (fig. 7). Therefore, the rear of the body will separate and the forces and moments will depart from the non-viscous theory and resort to experiments are mandatory.

Therefore, in order to understand the practical behavior of airplane one must understand the fundamental behavior of flow fields locally and in space. One must think of flow fields in terms of interference effects from various airplane components. The best illustration that I know of is the research that led to the "area-rule" concept. The area-rule concept was the direct result of experimental research designed to understand the interference flow fields of wing-body combinations in the transonic range. The development and verification of this concept was the result of transonic wind-tunnel studies such as pressure distribution, schlieren, tufts, and space measurements. Analysis of these results leads to an understanding of transonic interference effects and quickly led to practical application. This was an excellent example of doing the "right research at the right time". The detailed application of the area rule was immediately applied to specific airplane designs. Research emphasis was placed on specific application of the "area rule" to such designs as the F-102, and F8U-1. As a result, the "area rule" research developed concept was quickly applied and the details so necessary to practical application were exploited.

Application of fundamentals - A research program is usually the outgrowth of problems either actually encountered or which are predicted on the basis of existing knowledge. It evolves after a series of conferences with those involved or those likely to be involved and contains the knowledge and ideas of

many. These discussions usually define fairly clearly the scope of the research required. The greatest role of research planning is to shorten the time between the conception of a basic idea and its development into a practical asset. This always requires the cooperative effort of scientist, engineer, and manufacturer. At this stage, we can assume that we will be able to define the problem fairly clearly; exactly what it is, what it results from, what factors are likely to affect it, and what difficulties it is likely to cause. It should be stated at this point that throughout the course of a research program coordination should also be maintained between the facilities involved. This can prevent much duplication of effort and the corresponding waste of time and money.

Decisions must be made on the importance of the variables involved in the solution of the problem. These decisions must be weighed with respect to the difficulty of achievement and their effect on the final results. The measurements necessary to define these variables must be known as well as means of obtaining them. Considerable effort is usually required toward designing and constructing the actual hardware with which the studies will be made.

Initial work, particularly in relatively unexplored fields, is usually accomplished using the simplest possible apparatus. Results of these studies will usually indicate a direction toward which maximum effort should be exerted. The process is then one of applying existing knowledge to future work. This in turn leads to the investigation of a continuing chain of new ideas.

The philosophy of wind-tunnel programming and procedure for testing is one of intelligent flexibility. Overall objectives are set forth by virtue of past experience or by operational defects observed or anticipated in prototype flight

studies. By frequent data evaluation in the course of the program the goal of the basic objective is reached, but frequently off shoots of information have had such an important impact as to become the real key to open a new field of study. In other words, a pre-determined program, rigorously carried forward to the letter is considered an unacceptable procedure and rarely achieves a satisfactory goal from the research point of view. The development of future aircraft operating well into the supersonic range will be a developed product whose design will be dictated by well planned supersonic wind-tunnel research.

Proper interpretation of the results of each study is of utmost importance. These results must be studied first from the standpoint of understanding the nature of the phenomena. Secondly, they must be examined to see what needs further study. Finally, all information must be extracted which leads to a correct application to a complete airplane. However, throughout the development of aircraft, certain research problems have been firmly established for the development of transonic and supersonic aircraft regardless of their mission.

Once the manufacturer for example, decides upon the general configuration of an airplane to meet contractual specifications, his design estimates must be substantial by extensive wind-tunnel testing of a scaled model. The wind-tunnel tests must cover the complete speed range in which the proposed aircraft is expected to operate and involves the determination of the performance, propulsion systems, stability and control, flutter, and buffeting.

PERFORMANCE

In 1952, Whitcomb gave us the key to the understanding why some arrangements have less wave drag than others. This he demonstrated experimentally in the so called "area rule". In a brief period of time, this concept was incorporated into every transonic design and truly resulted in a design philosophy.

The effect of the application of area-rule principles to an airplane is summarized in figure 8. The area diagram of the prototype illustrates clearly the building-up of the area elements corresponding to the area development. The prototype area developments are much improved by "waisting", elongating the fuselage, such that the wave drag was substantially reduced. Therefore, the equivalent body concept is of great importance to the designer. Analytically the wave drag which occurs at transonic and supersonic speeds can be evaluated to a reasonable degree theoretically by evaluating the von Karman wave-drag equation. Evaluation of this equation has been done using Fourier series computations. This theoretical development has been extremely useful to the experimenter in guiding his research programs because he can, prior to his experiment, obtain some indication of the magnitude of the changes in wave drag associated with different configurations. However, the use of these theoretical methods requires (at least the Fourier series analysis) the use of analog computing machinery to obtain the results in a reasonable length of time.

We have to admit that at the present time we have very little background material on which to base the drag of a full-scale transonic airplane. All of the drag estimates that can be made at supersonic speeds are based on wind-tunnel and rocket model results. However, there seems to be no reason to doubt that wind tunnels do not give the correct wave drag of a model which can be applied directly to the full-scale airplane, IF,

the skin friction drag can be separated. There is a possible source of error in the estimation of supersonic wave drag if the displacement thickness of the boundary layer is of the same order of magnitude as the thickness dimensions of the model. This is the case of small models in high density supersonic tunnels. It is simply that in this case a variation in displacement thickness will amount to a variation in thickness ratio.

The interference problems arising from the use of propulsive jet exhausts from the rear of a body or nacelle have been recognized for some time. Although some experimental work has been accomplished, knowledge at the present time of the effects of a propulsive jet on the aerodynamic characteristics of a body from which it is issuing is still quite limited, particularly in the transonic and supersonic speed range. It can be anticipated that the effect of the jet would be to alter pressure over the rear portion of a body resulting in changes in base pressure and even to a minor extent skin-friction drag. Factors which influence the effects of the jet are free-stream Mach number, afterbody shape, base area ratio, jet velocity ratio, jet temperature ratio and jet pressure ratio. These factors all exert an influence at the base annulus. The jet pressure ratio is the primary variable which controls the size and general shape of the jet as it issues from the nozzle. Jet velocity is important since the difference between the jet velocity and the local stream velocity determines the amount of shear, and, therefore, the mixing since within the jet the viscosity characteristics of the flow are affected by the temperature. The jet velocity combined with the jet temperature is, therefore, a factor which is involved with the aspirating effects of the jet-mixing characteristics. The effect of the specific heat ratio is involved with determining the slope of the disturbance

lines at the beginning of jet boundary and also the shock wave diamond patterns within the jet. Of course, a primary variable is the variation of the exit geometry, particularly the nozzle shape.

In the past, it was frequently assumed that the jet characteristics were determined mainly by the jet momentum regardless of the temperature and other characteristics. Obviously, with regard to base pressure and the general pressure and flows near the base such an assumption cannot be valid. The general size and shape of the jet including the shock wave diamond pattern within the jet and the corresponding external pattern are clearly determined by the pressure ratio and by the specific heat ratio; also, of course, by the nozzle design if the exit is not a simple contraction. Furthermore, the very important mixing effects at the boundary of the jet which are of fundamental importance in determining the local pressures are affected by the local velocities or Mach numbers and by the viscosity and thus depend on temperature, in addition.

At the present time, it is difficult to estimate the base drag not only because of the lack of data and because the exact exit configurations and operating conditions have to be duplicated. In most wind-tunnel tests, the base drag is measured and subtracted from the drag because it is known that the base pressure existing on the model without engines is not the same as it would be for the airplane with engines. The base drag of the nacelles on the airplane must, therefore, be estimated and either added to the drag or subtracted from thrust.

Based on our experience, it is felt that for tests of complete models at Reynolds numbers of less than 10×10^6 , the transition point should be fixed, although this condition means inaccuracy in the skin-friction drag itself due to fixed transi-

tion. On the basis of the foregoing discussion, certain wind-tunnel and rocket-model tests may be set up to obtain basic data to estimate full-scale characteristics. This discussion will be limited to Mach numbers in the transonic and supersonic range and will not apply to the subsonic range where the evaluation of aerodynamic drag is well documented. Wind-tunnel tests to determine the lift-drag polars should be made with transition point fixed at the average full-scale location. This location is generally selected as 10 percent of the cord. For example, the wave of the polars will be defined by these data; that is, the incremental drag coefficient above the zero-lift drag coefficient will apply to the full-scale airplane. Tests of an equivalent planar wing with transition will determine the camber effects on the zero lift drag. It is desirable that further tests of a model with transition free at high Reynolds number at high lift (that is, rocket model data on wind-tunnel data with transition fixed corrected for the drag of the boundary-layer trip or high Reynolds number data transition free) will give a model zero-lift drag. The zero-lift drag thus determined from these data can then be corrected for Reynolds number by using the turbulent skin-friction curve for a flat plate. This is done by the usual method of obtaining individual component skin-friction coefficients by determining individual Reynolds numbers and weighted, wetted surface areas.

The weighted average of the lengths in the stream direction are used for Reynolds number determination and for most cases you will find that this corresponds closely to the mean-aerodynamic chord of the wing. The transition strip for most all model tests has a low drag estimated to be 0.0005 in wing drag coefficient. This transition strip is usually 0.1 inch in width with N° 120 carborundum (0.02" - 0.03" high) approximately 30-40 to a measured inch (see ref. 3). It is found using these

techniques that the model skin friction will follow very closely to the turbulent flat plate skin-friction drag and are approximately 6 percent - 10 percent higher, depending, of course, on the model surface conditions. This difference is attributed to induced velocity, separation, and roughness. In fact, the higher smooth model skin-friction drag can be accounted for by assuming only a 5 to 6 percent increase in velocity.

The accuracy of extrapolating the model skin-friction drag to full scale depends entirely on the characteristics of the airplane. It is known that depending upon the roughness of the full-scale surface a critical value of Reynolds number can be reached where the skin friction will remain essentially constant. This value of critical Reynolds number becomes smaller as the roughness is increased. At the present time, it appears that the turbulent flat plate skin friction curve predicts a lower full-scale skin friction curve than that obtained from full-scale airplanes and this is attributed to critical Reynolds number effects caused by manufacturing irregularities and leakage. Therefore, in order to insure the lowest possible drag it is of utmost importance that manufacturing techniques be cognizant of these effects.

PROPULSION SYSTEMS

With the rapid advance in the field of jet propulsion, it is necessary to allocate a considerable amount of research effort to the solution of the problems associated with the installation of these systems. These systems can be subdivided into three main categories, i.e., the inlet, the engine, and the exhaust system. Fortunately, it is possible to study these elements separately.

Inlets - Considerable effort was exerted at an early date toward solution of some of the basic problems related to air inlets at both transonic and moderate supersonic speeds. Early research was directed along two different approaches. In the first, the primary objective was to provide high pressure recovery with good flow distribution to the turbine engine at transonic and low supersonic speeds for a wide range of engine operation conditions. In the latter, the effort was directed towards obtaining low external drag with good flow stability over a small range of engine operating conditions at moderate fixed supersonic speeds for a ramjet engine.

Improvements in materials and methods available for applications to propulsion devices soon led to the realization that turbine engines and other potential types of air breathing engines could be operated well into the moderate to high supersonic speed range. As a result, the air intake system is required to satisfy a greater number of requirements which would make the selection of the basic design significantly more difficult, and the optimization of the components now requires increased research effort for each individual application.

Consideration of the inlet must require a high value of thrust minus drag, which necessitates a study and compromise of both internal pressure recovery and external drag. The inlet

must be matched to the engine operating requirements, and it must deliver high energy air to the engine with a minimum flow distortion and without pressure fluctuations for a wide range of air-flow conditions. Although basic research effort can be directed at improving isolated components, the aerodynamic interdependency of the intake system and the entire airframe preclude its separate study in the final analysis. Models for testing in the wind tunnel must in general, therefore, incorporate the correct geometric inlet configuration and provision must be made for its operation near the proper internally required mass flows so as to insure correct external flow in the field of influence.

The thrust-minus-drag problem is of course in reality two problems tied together inasmuch as each lends itself to some independent investigation. The thrust problem, insofar as the air intake system is concerned, is directly involved with total pressure recovery since both engine mass flow and jet thrust per unit mass flow vary directly with pressure recovery. The importance of high pressure recovery is reflected by the value of the change in net thrust per unit change in pressure recovery stated in percent. Thus, as much as 1.5 percent increase in thrust may result from an intake recovery increase of only 1.0 percent. On the basis of thrust-minus-drag, this also means that even if the external drag were to increase by 1 percent to achieve the 1 percent increase in pressure recovery that a net increase in thrust-minus-drag of 0.5 percent would still result for the case selected.

The drag problem breaks itself down into several categories: the minimum drag of the inlet at design operating condition, the drag increase at off-design conditions, and the drag or thrust associated with the incorporation of the intake system into the entire vehicle. For minimum drag of the inlet itself, the internal-contraction type with a stove pipe or straight cylinder

externally immediately suggests itself. This may prove to be correct if the capture area of the inlet is as great as the maximum engine cowling required, a solution not generally expected. Some cowl drag may therefore be generally required to be accepted unless the inlet can be incorporated into the wing or fuselage, in which case the bookkeeping is even more complex. Because the obtainance of high internal recovery may intail some form of boundary-layer control, the drag associated with this must be included with the cowl drag to obtain the minimum drag at design conditions. At off-design conditions, additive drag or by-pass drag will probably result depending on whether the excess of air is handled externally or internally. Finally, consideration of the entire airframe may suggest either a positive or a negative interference drag chargeable to the intake system. Considerable research effort is required in each of these problem areas, which in some cases may be studied with the aid of simplified models, but in general require significant detailing of the models and measurement techniques.

The increased importance of maximizing the thrust-minus-drag parameter, especially at high supersonic speeds, has resulted in increased research efforts directed especially toward obtaining high pressure recovery, but with due regard for the external drag problems. The supersonic airplane must operate a considerable portion of its flight time at "off design" operating conditions. By "off-design" is meant not only Mach numbers other than the maximum design values, but at mass-flow ratios significantly different from design values, and at angles of attack and/or yaw other than optimum values. A study of these variables must be made in the research facility to insure that engine flame-out, severe reductions in thrust-minus-drag, or intermittent stalling and/or surging of the engine, which may lead to mechanical failure, will not occur. Usually some compromises may be expected to result from alleviating some of

these difficulties. The problems may arise from engine-intake mismatch, lack of adequate variable geometry provisions, improper boundary-layer control, deterioration of engine-face velocity profiles, or low and/or high frequency pressure fluctuations in the intake system.

In summary, the role of aeronautical research with regard to intake design is:

1. To study new and improved methods for obtaining efficient compression of the air.
2. To determine and reduce the drag of the intake system itself and when it is incorporated into the entire airframe.
3. To ascertain and improve the Mach number, angle of attack and/or yaw, and stable mass-flow range.
4. To improve or reduce the flow distortion and pressure oscillations at the engine face.

It should be noted at this point that in the wind-tunnel testing of specific models containing air inlets that the correct operation of these inlets is essential to the measurement of correct forces on the model. By this is meant that the inlet should operate at proper mass-flow-ratio values in order to insure proper external flows over the model. The actual internal drag of the model duct is not considered a problem since methods are available for reducing the data to a datum which excludes this force.

Engine - While a particular engine is usually the result of research and development, there are several problem areas which require separate research effort and may be classified as basic research. It is, of course, essential that the inlet supply the engine with the necessary mass of air at the maximum

possible pressure. Further, the air must be ducted from the inlet to the engine compressor face with the minimum losses and at a distortion level which the engine can accept.

The turbine engine may be broken down for research study into its three principal components: the compressor, the combustors, and the turbine. The ramjet engine, because it lacks both compressor and turbine, may be studied as a combustor only. In the compressor stage, the problems of surge and stall, compression ratio, and efficiency are undergoing continuous aerodynamic research both experimentally and theoretically. Similarly, in the combustors, studies of fuel-oxidizer mixing, ignition, flame holding and propagation, and combustion efficiency are being made to effect improvements in specific thrust and fuel consumption. With the turbine section, the handling of the gases over a wide range of temperatures and pressures at increased efficiencies has received extensive study.

In addition to the aerodynamic and thermodynamic problem areas suggested, the associated fields of fuels, vibration and flutter, and materials research should be mentioned. In fact, many of the present limitations imposed on current types of engines are directly associated with these last mentioned problem areas, for instance, the materials for operation at high temperatures.

Finally, after the manufacturer has made use of all available research information and the necessary development has resulted in a complete engine, overall performance must be established in the research facility. Additional aerodynamic information is required to assist the inlet design such as the distortion limits under which the engine can operate satisfactorily and the flow fluctuations which can be tolerated.

Other components of the internal system can be developed

largely on the basis of generalized engine requirements. It is necessary to have a knowledge of the corrected air flow and fuel flow, inlet and outlet temperatures and limit, and operating pressure ratios. The design of the exhaust system essentially requires specific knowledge of the engine outlet conditions.

Jet exit - In the jet-exit field as in other fields the existence of a potential problem was predictable. Rapid advances in the use of jet propulsion were foreseeable and the supply of existing information was inadequate to cope with the problem. As a result, a research program was formulated to define the nature of the problem.

Initial work in this field was aimed at determining the relative importance of the many variables involved. Relatively simple models were used in these original investigations. These tests began to clarify the problem. In addition to giving some insight into the phenomena some information of use to the airplane designer was obtained. These tests also indicated regions requiring further investigation.

As an example, the original investigations indicated that the afterbody of an airplane fuselage or nacelle should be faired in such a way that no flow separation existed and so that the base annulus surrounding the jet was at a minimum. By doing so, it appeared that the jet would have a favorable effect, i.e. a drag reduction. It was further indicated that regions of separated flows resulting from the use of blunt shapes or large base annulus sizes could result in severe adverse effects. These tests also established jet pressure ratio as a primary variable and indicated the effects of jet temperature. It was found that the effects of temperature were small or negligible for low drag shapes while for the blunt shapes the effect was

found to be sizable. Additional tests further clarified and verified the predictions made from the first results. Valuable knowledge as far as testing techniques was also obtained.

Many areas of research are inter-dependent and tend to progress together. Along with the answers to external flow problems being answered above the development of internal flow configuration was also proceeding. As flight speeds increased and correspondingly engine-operating pressure ratios there also came a need for cooling flows due to the high temperatures encountered in tailpipe afterburning. These developments led from the simple tailpipe with a sonic nozzle exit to the variable convergent-divergent ejector. The all-variable ejector, of course, resulting from the fact that a fixed convergent-divergent passage can only be designed for one condition while off-design penalties may be severe. The necessity for these variable exits has presented in turn a problem in afterbody design in that while a satisfactory fairing may be achieved for the nozzle in the full open or maximum speed condition, off-design operation, which requires the closing of the exit, produces a base annulus region which is a potential source of drag.

Here then the designer of an airplane is faced with a choice. He can use a completely variable exit with its greater weight and the problem just described in order to obtain maximum thrust or he can compromise on a fixed ejector which allows a good external fairing and hope that the saving in drag will compensate the loss in thrust. This is a major problem in high-speed aircraft.

Figure 9 is an example of the exit situation. Here, the thrust ratio for four ejector configurations is shown against Mach number. This ratio is the relationship of the thrust obtained from the ejector to that which could be obtained were

the primary nozzle flow expanded isentropically to ambient conditions. Curves designated 1-4 (fig. 9) are for afterburning conditions where primary jet temperature is 3500°R . Here it is indicated that a fixed configuration can be designed for optimum operation at only one speed and suffers from off-design operation. However, a completely variable configuration could be made to operate along the peak thrust line. The complications of such a variable device are obvious. In the lower part of the figure, exit N° 3 has been modified for the non-afterburning case (1600°R , primary nozzle closed down) as configuration 3(a). Here, severe losses occur; thrust ratio is far below the cylindrical ejector (§1) for example. This indicates the nature of the problem facing the designer, i.e., the use of a heavy, complicated variable shroud or a compromise fixed ejector and an improved external design.

While it is questionable that results from small-scale models can be applied exactly to a full-scale article, particularly those obtained on simple models, this information tells the designer which way to go. He is thus able to design an unique configuration which may not be a perfect optimum but will not be too far off so that a correction does not take the form of a major modification.

It should be pointed out that it is now possible to test almost exact powered models of proposed configurations employing jet propulsion systems. Thus, corrections can be made at much less time and expense.

Previous discussion has been confined to jet-exit difficulties with the body from which it issues. The presence of a jet presents other problems also. For example, the interference effects on adjacent bodies or lifting surfaces.

Figures 10, 11, 12 and 13. These tests were conducted in

the 9-inch tunnel using a cold jet exhausting from a nacelle placed at various locations with respect to the wing. A drag breakdown for the central body as well as schlieren photographs of the flow field is presented. Shown are two typical examples. The important point here is that jet interference effects may increase or decrease the fuselage drag depending on the nacelle location and operating pressure ratio. The fuselage drag variations due to jet interference may be of the same order of magnitude as the changes due to addition of the nacelle or due to varying the nacelle location with the jet off. These results indicate that it is not sufficient to consider only the location of the exit shock and the shock within the jet in the analysis of fuselage drag; rather, the entire jet-interference flow field must be considered.

Let us now examine the effect of jet exhausting under a wing. Figures 14 and 15. Here, a sonic nozzle jet has been exhausted from a nacelle placed at various heights and spanwise stations beneath a wing. In the upper portion of the figure are some typical data obtained that show the increment in pressure coefficients resulting from operation of the jet. In the lower portion is shown the changes in spanwise loading resulting from these effects.

It is seen that the largest effects occur at a Mach number of 1.05. The reason for this will be quite clear when referring to figure 16 which shows jet flow phenomena. Here, it can be seen that for the subsonic case, the internal disturbances within the jet are reflected from the subsonic mixing boundary while for the supersonic case these disturbances penetrate the boundary to extend into the free stream and consequently effect any adjacent surface or body to a greater extent. Some of these effects can be quite large. Obviously, the jet stream can influence the flow about these other components with

resulting changes in forces and moments. Here again the approach has been to utilize relatively simple models to obtain guidance information and to proceed then to more precise evaluations. These studies have indicated desirable locations for nacelles and tail surfaces. Unfortunately, until more is understood about such things as jet mixing and displacement effects, these studies can only serve as a guide. The exact evaluation of a unique installation still requires individual examination.

Many additional problems exist in the field of jet exits. Among these are the use of jets for aircraft control, particularly in the field of VTO aircraft. Here mechanical as well as aerodynamic problems exist. Either the exit nozzle must be swiveled or means must be found to deflect the jet at a response rate adequate for this purpose. While much exploratory work has been done and VTO aircraft are actually flying, much improvement is needed.

One of the most pressing problems at present, particularly to the designer of commercial carriers, is a means of suppressing the noise level of jet propulsion systems to acceptable levels. While some quick fixes are being attempted much effort is being devoted to understanding the nature of the noise. Attempts so far to quiet the noise have resulted in thrust losses, weight penalties, and shapes difficult to handle from the standpoint of external aerodynamics. A satisfactory solution has not been forthcoming.

Figure 17 gives a brief example of some of the work being done towards suppression of jet noise. This slide shows the change in sound power ratio as a function of Lighthill's parameter. Here P_{std} is the sound power level of a standard nozzle. It has been established (TN 3974, ref. 4) that sound

power radiated from a convergent nozzle jet can be correlated using Lighthill's parameter and is a linear relationship. Thrust losses show up as a decrease in velocity and a consequent decrease in this parameter.

From the slide, it can be seen that increasing the number of nozzle segments, and consequently breaking the mixing into smaller vortices, tends to lower the sound power level. Here it should be noted that while the 12-segment nozzle appears to be reasonably good sound suppressor, it also caused a thrust loss of as much as 7 percent. Difficulties with external fairing as well as added weight are not difficult to predict.

Another problem, which may have a common solution from a hardware standpoint with noise suppression, is that of thrust reversal. Full or partial thrust reversal is desirable under certain flight conditions as well as in landing. This is particularly true for the wave-off condition where it is desirable to have rapid control over forward thrust. Many devices have been considered to date with varying degrees of success. Uniformly, however, these devices have been bulky, causing storage difficulties as well as weight problems. In addition, some have caused severe buffeting when extended in flight. More effort is necessary.

STABILITY AND CONTROL

Modern trends in the development of aircraft and missiles have radically affected the analysis of wind-tunnel experiments and also flight testing with regard to stability and control. In the past, wind-tunnel and flight test analyses were concentrated on the performance and static stability characteristics. Today, we find that the modern aircraft depends not only on setting certain static stability and performance characteristics but also on their possessing suitable dynamic properties. As a result, increasing emphasis is being put on experimental and analytical stability analysis in providing new experimental techniques and the use of digital and analog computing equipment. The modern trend in aircraft is the utilization of automatically controlled equipment and feed-back systems for the human pilot to augment the damping oscillations about one or more of the principal axis of the airplane. The modern airplane is confronted with problems ranging from the low-speed stall through the difficult transonic speed range to supersonic speeds as well as altitudes from sea level to well over 50,000 feet, not to mention load factors ranging from negative limits to positive limits. In the past, with regard to dynamic stability, the principal objective was to determine the time to damp the motions to half amplitude, and the number of cycles required following disturbances from level flight conditions. The modern aircraft are capable of severe maneuvers for both piloted and automatically controlled aircraft. Aerodynamically, the modern aircraft is involved with many configurations having such features as sweep-back, low aspect ratio, thin wings, and slender fuselages. Most of these configurations suffer from non-linear aerodynamic effects. Problems of separated flow and their consequences have plagued the airplane designers for years. The current trend in design has, therefore, aggravated problems of separated flow. It has been stated before that in order to evaluate research problems

one should try to have an understanding of the flow phenomena associated with the problem to be investigated. This, of course, applies to analytical as well as experimental studies. Therefore, in discussing stability and control, a few remarks pertinent to flow phenomena should be made. Analytical evaluation of the stability derivatives generally involves potential-flow solutions representing the wing by a vortex sheet, doublets or sources. The derivatives are obtained by integrations of the resultant forces. The same considerations can be given to the flow fields away from the wing and thereby one can derive the effect of the wing on other components. As we know, this is fine IF, we did not have boundary layer which violates the basic assumptions. Separation of the turbulent layer for thick wings and laminar separation for thin wings near the leading edge are common. At high speeds, shock-induced separation and the shock moves rearward with further increase in speed, leading-edge suction disappears and as a result the stability derivatives are critically dependent on Mach number.

The foregoing description is not complete but does illustrate that recourse to well planned experiments based on an understanding of flow phenomena is indicated. To obtain quantitative values one should bear in mind that the complex flow fields encountered should be duplicated and evaluated by the selection of the proper variables.

The body or fuselage can no longer be neglected. It is important with regard to stability and control and in many cases of equal importance to the wing. This is true simply because high speeds maneuvers with swept and low aspect ratio wings incline the fuselage to high angles. This results in low pressures over the forward portion of the fuselage. Thus, forces result that form a couple and result in a significant destabilizing moment. Of course, separation will also occur

and cross flows will develop which further complicate the force system. The separation near the base of the bodies will be evident by vortices that trail along the axis downstream and increase in magnitude as the angle of attack is increased. Now, since the wing intersects the body, the up-flow induced by the body will influence the presents of the wing and may alter this vortex flow depending on the location of the wing. The induced flow on the side of the body also influences the wing pressures. At supersonic speeds, the bow wave caused by the fuselage may effect the wing tip and last, but not least, the tail is affected by the Mach disturbances and vortex flow of the wing and body. Thus, the flow field will be such that the tail depending upon its location will be subject to different flow angularity and velocity.

The presentation of experimental data in a most convenient form is worthy of some consideration. In the beginning, wind-tunnel and flight tests techniques were concerned with the performance capabilities of the airplanes. These measurements required for performance analyses were made on what might be called a path axis system where the forces revolved along and perpendicular to the direction of the motion. Early dynamic analyses tended to follow these conventions. In the United States, this system is referred to as the stability axis system and wind-tunnel balances were designed to obtain measurements which could be converted to these axes with very little effort. In our modern transonic and supersonic wind tunnels, a sting support is used together with a strain-gage balance mounted inside the model which measures the forces and moments with relation to the axes fixed in the model. It has been the common practice to convert these data to the stability axes before publishing them. The analysis of large-scale maneuvers, however, requires substantial changes in angle of attack and sideslip and the stability axis system is not convenient simply

because they cannot be considered fixed. As a result, the body axis system is more appropriate, particularly since the flight test instrumentation is also referred to in this manner. In other words, data obtained from internal strain-gage balances are already in the most useful form for large-scale dynamic studies and need not be converted to the traditional stability axes.

Static stability - In order to reduce "Compressibility effects", airplane configurations employ thin wings, sweep, and low aspect ratio. However, in going from subsonic to supersonic speeds the center of pressure of such wings move rearward from 10 to 25 percent of the chord depending upon the planform. As a result, at low supersonic speeds, the stability may become excessive. A good indication of many longitudinal stability problems is the lift-curve slope. Figure 18 illustrates the effect of thickness and sweep on this parameter.

The problem of achieving acceptable stability or the desired variation of pitching moment with lift is troublesome. The linear variation of pitching moment with lift with an increase in stability at high lifts is generally considered the most desirable. In order to guide designers it was found that the high-lift pitching-moment characteristics of wings depend primarily on the planform that is, aspect ratio and sweepback. This was accomplished by analysing wind-tunnel results which are illustrated in figure 19. Therefore, a boundary was established, separating the stable sweep-aspect ratio combinations from the unstable. In determining these first order effects the designer has an important design criterion to work with.

Low-speed research continued with effort being placed on the understanding of the flow phenomena, and developing means to improve the stability. The work was also extended to include studies with tails as a logical step. As a result, a large

amount of this work was summarized by Furlong and McHugh. Here then, we have a rather complete picture of the low-speed characteristics of wings for high-speed aircraft.

Tip stalling of swept wings or triangular wings impose some problems with regard to tail location. The stalling of the flow causes the inward deflection of the tip vortex resulting in downwash changes that influences erratically the contribution of the tail to stability. The designers ability therefore, to solve his stability problems, becomes one of matching the nonlinear contributions of the wing-fuselage combination with the tail contribution to provide a desirable result.

As was previously mentioned, another problem area which plagues the airplane designer is the increased stability of the airplane as the flight speed is increased from subsonic values to low supersonic values. This excessive stability is a result of several generally well-known factors. These factors include the increase in stability of the wing-body combination that is caused by a rearward shift of the wing center of pressure and a stabilizing interference effect of the wing lift carried over to the afterbody. The stability is further increased because of the loss of the subsonic type of wing downwash at the tail since the major portion of this downwash is confined to the wing-tip Mach cones and at supersonic speeds begins to move off of the horizontal tail. In addition, in the case of most low-tail airplanes, stabilizing upwash from the body may be encountered. At the same time that the stability is increased, the effectiveness of the tail in producing pitching-moment is reduced. These effects combine to cause large untrimmed pitching moments that must be overcome through rather large control deflections, and the result is high trim drag and low trim lift-drag ratios. In addition, because of the large control deflections required for trimming, little excess control deflection may be available

for maneuvering.

High-speed pitchup - The primary solution to the pitch-up problem is the elimination of the basic trouble, the nonlinearity in the static pitching moment. There are two general lines of attack. The first is the application of "fixes" to wings, as in the form of fences, for example; the second and more fruitful is the positioning of the horizontal tail vertically such that it remains in or moves into a favorable gradient of downwash with angle of attack, moving into a decreasing rather than an increasing downwash field. Practically, combinations of both fixes probably lead to the best results.

Thus, it appears that airplane characteristics which can cause pitchup can be recognized readily from wind-tunnel data. Motion calculations using these wind-tunnel data, however, should be made to indicate the severity and danger of such characteristics for any given design. As an example, the static pitching-moment nonlinearities observed from wind-tunnel tests of a model of a 60° sweptback airplane were relatively mild when compared with some of the instabilities noted for various other swept-back wing model configurations. From such a casual inspection of the static pitching moments, it might be expected that pitch-up would not be particularly severe; however, the airplane was found to have marked pitch-up during flight tests. Dynamic response calculations were therefore made in order to determine the true significance of the pitching-moment nonlinearities. These calculations based on the wind-tunnel data predicted a pitch-up motion of the airplane that was in good agreement with flight results.

DIRECTIONAL AND LATERAL

Directional stability of an aircraft is the most difficult to predict because it is influenced by so many different factors. As a very first approximation what is usually considered adequate is simply the evaluation of the lift-curve slope of the vertical tail. Therefore, the effect of Mach number is the first concern. Thus, the directional stability derivative, C_{n_r} , will have a variation with Mach number, if not influenced by other factors, similar to the variation of lift-curve slope. Thus, one can easily predict on this basis that at supersonic speeds the configuration will have a deficiency in directional stability and will dictate the size of tail required. At higher angles of attack, our directional problem becomes increasingly difficult. For example, separated vortices from the fuselage can have serious effects on the vertical tail. The height of the wing with relation to the fuselage (high, mid, or low position) will effect the sidewash flow at the tail.

The effective dihedral derivative, $C_{2\beta}$, is obviously critically dependent upon the flow characteristics over the wing. Unsymmetrical flow separation makes this parameter non-linear. Sweptback wings whose separation occurs at the tips will make this parameter to become abruptly positive at relatively small angles of attack. It should be noted that this is in contrast to straight wings where the flow separates first in the root sections causing an increase in effective dihedral.

Experiments to determine the effects of oscillation frequency on the combined damping in yaw derivative, $(C_{n_r} - C_{n\dot{\beta}})$ have indicated negligible variations within the range of frequencies normally applicable to airplane motions. However, this wind-tunnel technique developed to obtain this type of data at transonic speed is new. This subject at transonic and supersonic speeds needs further investigation.

CROSS-COUPLING ENCOUNTERED IN LARGE-SCALE MANEUVERS

Dynamic stability - The high-speed dynamic stability problems first became apparent when World War II airplanes met catastrophe apparently because of a divergent pitching motion in the transonic range. This instability we know was caused by the strong shock-induced separation that occurred on the relatively thick wings. Thin wings reduced this problem, but the problem still exists and for tailless configurations should not be dismissed. To illustrate, figure 20 compares the results of a delta wing-body combination experimental one-degree-of-freedom with wing-body variation with Mach number. It is encouraging to note the excellent agreement with theory in that this reversal in damping is predicted. To fill out this picture, it becomes obvious that further investigation would be needed to examine the parameters separately. The point illustrates the importance of defining the entire contribution of the variables and not being misled by just looking at one. Since most airplanes have tails, the wing damping whether positive or negative is small compared to the contribution of the tail.

In large-scale maneuvers of modern aircraft, the lateral and longitudinal responses of the aircraft cannot be considered independent since cross-coupled effects are greatly influenced by the inertial distribution found in modern aircraft. Essentially, the modern aircraft is different from what might be considered a standard subsonic aircraft in which the mass is distributed more or less equally along the fuselage and wing span resulting in approximately equal moments of inertia about the aircraft axes. We now have configurations where the moments of inertia in roll, for example, are an order of magnitude smaller than the moments of inertia in pitch and yaw. The effect of this trend is one that results in inertial reactions and cross-couples forces to dominate over aerodynamic stiffness

and damping forces. The problem, therefore, is divided into two parts; one deals with the mechanical inertia forces, together with the aerodynamic forces. Illustrated in figure 21 is a model that is rolling about its flight path. The mass along the fuselage is considered in two parts; fore and aft of the center of gravity. The resulting centrifugal force will produce a pitching moment that is a function of the angle of attack and the rolling velocity. Resisting this moment is the longitudinal stability of the airplane. Thus, it is obvious that if the rolling velocity is high enough, the resulting centrifugal forces can overcome the inherent aerodynamic stability and the airplane will encounter whirling divergence. Now let us consider the case where the airplane is rolling about its principal axis illustrated in the next figure, figure 22. For this case, the centrifugal forces are zero and when the airplane rolls to 90 degrees the angle of attack has now become the sideslip angle. The aerodynamic stability will now introduce restoring moments both in yaw and pitch and make the airplane (if the roll rate is high enough) initiate a "whirling divergence".

These simplified illustrations of the cross-coupling effects are indicative of the parameters that need evaluating. Therefore, in order to properly evaluate the behavior of modern aircraft, it is necessary not only to examine the static and dynamic derivatives but such derivatives as pitching moment versus angle of sideslip, yawing moment versus rate of sideslip, and other cross-coupled derivatives such as yawing moments due to aileron deflection. To the experimenter, therefore, in his analytical analysis, he is confronted with the equation of motion becoming non-linear and must in his analysis consider both longitudinal and lateral characteristics simultaneously to obtain realistic results. As a result, the experimenter is confronted with the problem of considering not three equations of motion but at

least five. In general, the areas where difficulty can be expected in the flight spectrum are where stability defects are encountered. For example, low speed near the stall and high altitude maneuver conditions.

In summary, we have seen that recent basic changes in airplane configuration, primarily the trend to sweep and low-aspect-ratio wings and the concentration of mass in the fuselage, have caused the general characteristics of airplanes to change such that dangers of large and possibly violent motions triggered by causes other than the stall itself are now possible. Of major importance are the pitch-up, directional divergence, the effects caused by inertia coupling during combined lateral-longitudinal motions, and reductions in roll damping and dihedral effectiveness at angles of attack below the stall. Wind-tunnel measurements may be used to recognize the existence of most of these problems, and in general, there appear to be ways of improving such deficiencies. The problem existing in the combined maneuvers, however, cannot be recognized directly from wind-tunnel measurements and will require some rather extensive calculations from free-flight rocket models to evaluate any given design.

FLUTTER AND BUFFETING

It is becoming more and more difficult for the airplane designer to avoid flutter while at the same time holding the structural weight low enough to meet ever increasing performance requirements. The way in which research experiments are used to help solve this flutter design problem will be discussed.

We begin with a definition. Flutter is conventionally defined as a self-excited oscillation resulting from a combination of inertia, elastic, oscillatory aerodynamic, damping, and temperature forces. In combination these forces can result in unstable motion (that is, flutter) which leads to marked or extremely severe structural failures.

The great majority of flight flutter incidents have involved the stabilizing and control surfaces rather than the wing itself. In recent years, in fact, wing flutter has occurred only on wings with external stores attached. Since the time that airplanes began to fly at transonic speeds, about half of all flight flutter incidents have been due to control surface buzz. This is a single-degree-of-freedom instability that occurs as a result of aerodynamic forces on flap-type control surfaces at transonic speeds.

There are numerous possible modes of control surface flutter. Flutter has occurred due to coupling between stabilizer bending and control surface rotation, between stabilizer bending and stabilizer torsion, between stabilizer torsion and control surface rotation, and between fuselage bending and control surface rotation. Here the word stabilizer is used to include both the horizontal and the vertical stabilizing surfaces. These all-movable stabilizing surfaces have also fluttered in flight and the possible modes include those involving control

actuator stiffness and fuselage bending and torsion as well as the vibration modes of the isolated stabilizer. The T-tail adds further complications because it introduces, in addition to the modes that have been mentioned, the possibility of strong coupling between the vertical and horizontal surfaces themselves.

If an automatic pilot or automatic stabilization is used in an airplane, then the automatic pilot may be involved in a flutter mode. In addition to these cases of primary control surface flutter, flutter of tab surfaces has also been encountered in flight.

Before describing what has been done experimentally to study the phenomenon of flutter, it seems appropriate to review the present status of the problem of analytical prediction of flutter. As in almost all aerodynamic problems, analytical analysis is a prerequisite to experiment and is preferred whenever the analysis is known to give dependable results.

A flutter analysis can be divided into three separate steps. The first of these is the calculation of natural frequencies and mode shapes. Several methods for making such calculations are given in the textbooks (for example, refs. 5-7, which include bibliographies). These methods give reasonably acceptable results for conventional aircraft. Some difficulty has been experienced in predicting the vibration modes of low aspect ratio surfaces such as are used on the newer supersonic airplane, and the higher vibration modes of wings that include concentrated weights (stores) elastically suspended. These problems are treated in references 8-12.

The second step in the flutter analysis is the calculation of the oscillatory aerodynamic forces. In this connection, the two-dimensional linearized theory is essentially complete (see refs. 7 and 13, for example). The validity of this theory at

transonic speeds is, however, open to question. An even more serious limitation to the usefulness of the two-dimensional theory arises from the very low aspect ratio of supersonic airplanes. The aspect ratio is so low that strip analysis based on two-dimensional aerodynamics, which has been very successful in the past, is no longer adequate.

In the opinion of airplane designers, the three-dimensional theory for wings oscillating in an elastic mode is not in a satisfactory state. Much effort has been expended on lifting line methods but these have serious basic shortcomings. In particular, the aerodynamic center position is assumed in advance so that no account can be taken of the variation in aerodynamic center location with Mach number, which is an important factor in determining flutter behavior at transonic speeds. The lifting surface methods (refs. 14 and 15 for example) appear promising. Further experiments are needed to evaluate the accuracy of these methods at transonic and supersonic speeds. The calculations are lengthy but high-speed computing machinery promises to make lifting surface methods practical for design use.

At high supersonic speeds, the so-called piston theory (ref; 16) seems to work well. It accounts for non-linear effects that are known to be important at these speeds, such as the influence of the airfoil thickness distribution.

The phenomenon of control surface buzz deserves special comment. It is a transonic phenomenon that seems to be associated with the shock waves on the wing surface ahead of the control and as such it involves nonlinear aerodynamics. Because the air forces cannot be calculated, this troublesome problem, which accounts for about half of all flight flutter incidents, is beyond the reach of analysis at the present time.

The third step in the flutter analysis is the actual calculation of flutter velocities and frequencies. Although some improvements are to be desired, the available numerical techniques (refs. 5-7) are adequate, in general, for solving flutter equations.

Certain areas have been noted in which analytical techniques are inadequate. A natural question would be: Is flutter critical in any of these areas? Figure 23 will help to answer this question. The solid line in this figure is the flutter boundary for a particular control surface. Flutter would occur if the airplane with this surface were to fly on the cross-hatched side of the boundary. Now, we know that supersonic airplanes are not allowed to exceed some limiting value of dynamic pressure, q , at low altitude. A dashed line on this slide shows the variation of Mach number with altitude for a constant, q , of 550 lb. per square foot. The critical point from the standpoint of flutter is that point at which the margin between the limiting q line and the flutter boundary is smallest. This point is found at an altitude of 20,000 feet and a Mach number of about 0.9. These results are typical in that the critical flutter conditions usually appear in the transonic speed range. But this is the very region in which the analytical techniques are least dependable. Thus, experimental flutter research is particularly important at transonic speeds.

Some of the large transonic tunnels in the United States are cooled by a continuous exchange of air between the tunnel and the outside atmosphere. As a result, these tunnels operate at a constant stagnation pressure of one atmosphere. In order to demonstrate the effect of this limitation on experimental flutter research, the flutter boundary shown in figure 23 is replotted in figure 24 as a function of tunnel stagnation pressure, H , and stream Mach number, M . During a flutter test

in a constant-pressure wind-tunnel, the Mach number is increased until flutter occurs. When the model begins to flutter, the tunnel speed must be reduced rapidly in order to avoid destruction. The experiment gives only one point on the flutter boundary; the remainder of the curve can be established only by testing a number of different models or by testing a variable model in which some of the stiffness parameters are changed. This, of course, is extremely expensive and time-consuming. Moreover, unless one has the ingenuity to design a model that can be varied during the course of the tests, points to the right of the minimum value of the flutter boundary cannot be reached at all, because the flutter region cannot be traversed without destroying the model.

The obvious answer to these problems is a variable-density wind tunnel. With the ability to vary the total pressure a number of points with a given model design can be obtained, and thus the flutter boundary is established. Points to the right of the minimum value can also be obtained by starting the tunnel at low values of the pressure, thus avoiding flutter at the lower speeds, and once having reached a fixed Mach number, then increasing the pressure until flutter is encountered.

At the United States, the greatest part of our transonic experimental flutter testing has been done in the transonic blowdown tunnel, which is a variable-density tunnel designed and built with flutter testing requirements in mind. (Fig. 25. It should be noted that, in order to achieve reasonable figure proportions, the horizontal dimension shown here has been shortened to one-half of the actual dimension.) We have also wished many times that this transonic blowdown tunnel were square or rectangular in cross section; rather than octagonal, so that, for example, semispan models could be tested with a reflection plate. The blowdown feature of the tunnel has the

obvious advantage of eliminating problems of equipment damage due to model failure. For a blowdown tunnel, this slotted tunnel has the unique feature of having independent control of Mach number and air density. The tunnel air supply and tunnel pressure are controlled by quickly-responding valves connecting the tunnel to the pressure source. Tunnel total pressures of from approximately 1 to 5 atmospheres can be obtained. The Mach number in the tunnel is controlled by the use of a series of orifice plates located in the diffuser which provide, through choking, fixed values of the Mach number. The maximum Mach number attainable is approximately 1.4. In designing and building properly scaled dynamic models, it is usually rather difficult to obtain a sufficiently light model having the other required characteristics. High-pressure ranges of operation relieve this problem considerably. Another feature, which is significant with respect to flutter testing, is the use of an extended sting. Such an arrangement avoids the typical bow-wave reflection interference at low supersonic speeds created by fuselage bow waves.

This tunnel has been used for several years for both general research investigations and investigations of components of particular aircraft. In specific investigations, flutter problems have been defined and cured on a number of airplanes.

Flutter testing at transonic speeds has also been conducted in some of the other NACA transonic wind tunnels on a limited scale. Two other tunnels having the desired variable-density feature are the Langley 8-foot transonic pressure tunnel and the Langley 2- by 4-foot transonic flutter tunnel. The principal advantage in the use of these tunnels lies in the larger-size models. Even with these facilities, these are imposed rather stringent limitations on model complexity, and such tests are usually limited to component parts such as wings, tail

surfaces, etc. In order to provide for tests of complete models, the low-speed Langley 18-foot pressure tunnel is being converted to a variable-density, transonic pressure tunnel using freon gas. The reasons for using freon gas are basically two: (1) for a given power and Mach number, a higher density is obtained; and (2) because of the lower speed of sound, the scale values of the Mach number and the Froude number can be more nearly met simultaneously.

In the study of the flutter of complex airplane models, the simulation of multiple body freedoms in experimental testing is of prime importance in cases where it is necessary to reproduce the interaction of major components of the aircraft. In recognition of this, the NACA has undertaken the development of a towed-airplane model flutter-test technique (figs. 26, 27, and 28) which was chosen to circumvent the transonic support interference problem, and at the same time, provide all the body freedoms except longitudinal translation. In this technique an airplane model is flown at the end of a tow cable in the wind tunnel under auto-pilot control. The model is provided with a five-component auto-pilot. The angular displacement of the model with respect to the tow line produces corrective elevator and rudder deflections. An electrically-driven gyro produces aileron deflections which are proportional to the angle of bank. These controls all have motorized variable linkages which can be remotely controlled from outside the tunnel. In addition, rudder and elevator deflections which are proportional to the yawing and pitching velocities are provided by gyro-servo combinations through mixing linkages. With such complex models, it is desirable to permit flutter and then safely stop the motion. In this case, a flutter stopper consists of a gyro which provides additional damping to the oscillatory motion when locked to the vibrating wing spar. To date the model has

been successfully flown in the Langley 19-foot pressure tunnel, and to Mach numbers as high as 0.6 in the Langley 16-foot transonic tunnel.

It appears that the problems associated with extending this technique into the transonic range are not insurmountable. This technique is being described here primarily to illustrate the extreme complexity of some of the model techniques associated with flutter testing, although such techniques are not expected to replace the simpler forms of component testing which are providing a means for a much greater volume of test results. The tow-model technique appears to have use in the study of other aeroelastic phenomena involving airplane response, and since the airplane auto-pilot system constitutes a variable-stability airplane model, the technique may offer a wide area in the study of dynamic-stability problems.

One final factor involved with the flutter testing of models has been the effect of wind-tunnel turbulence. In all of the work involved with flutter testing at the Langley Laboratory, this has been a question of considerable interest. Briefly, the problem arises as the flutter speed is approached, because the damping of the system then approaches zero. In this range, excitation of vibratory modes akin to the flutter modes by tunnel turbulence can be mistaken for flutter. Two types of turbulence can cause such effects: (1) a generally high turbulence level; and (2) discrete high spikes in the turbulence frequency spectrum of a tunnel with relatively low overall turbulence. Such effects have led, in several instances, to the determination of erroneous flutter speeds and to uncertainties in the application of results. It is emphasized that considerable attention must be paid to these effects where flutter work is intended.

Buffeting - One objective of buffet research has been to learn how to predict flight buffet loads from wind-tunnel measurements. A method for meeting this objective is described in reference 17. Bending-moment measurements were made with electrical strain gages mounted near the wing root. By combining knowledge from two different fields, those of structural beam theory and statistical analysis, an equation was derived for predicting flight bending moments from the buffet measurements on the model.

Model damping requirement - One requirement for a buffet model concerns the damping of the buffet vibration. The question that arises is whether the damping is aerodynamic, structural, or some combination of the two. This question can be answered, in a given case, by studying the variation of buffet bending moment with air density (fig. 29). In order to facilitate comparison between different tests, the density values for a particular test have been normalized by dividing by the maximum density for that test. Similarly, the bending moments for a particular test have been normalized by dividing by the bending moment at the maximum density. The bending moment is herein defined as the root-mean-square value of the vibratory part of the wing-root bending moment; the static bending moment has been removed from the data.

If the damping is structural, the damping coefficient will be constant and the bending moment will be directly proportional to the exciting force and thus to the air density. This variation is shown by the dashed line in figure 29.

If the damping is aerodynamic, on the other hand, the damping coefficient and the exciting force will both be directly proportional to the density. By the methods of power spectral

analysis, the root-mean-square bending moment is found to be proportional to the square root of the density. (See ref.18). This variation is shown by the solid line in figure 29.

Flight-test data indicate that the damping of the buffet vibration is primarily aerodynamic (ref. 19). In order to illustrate this point, some flight results are plotted with circular symbols in figure 29. Such results led to the assumption of aerodynamic damping in developing the equation of reference 17 for scaling buffet bending moments from wind-tunnel models to full-scale airplanes. The equation is applicable, therefore, only when the damping is primarily aerodynamic for the model as well as for the airplane.

In order to determine whether the damping of models actually is aerodynamic, bending-moment data for three ordinary force-test models are plotted in figure 29. Model 1 is suitable for buffet loads tests because the damping is primarily aerodynamic. For models 2 and 3, on the other hand, the damping is largely structural; thus, the scaling equations of reference 17 do not apply to these models. The model data shown in figure 29 represent two extremes, and anything between these extremes is possible.

If the model is tested in a variable-density wind-tunnel, a plot like figure 29 provides a means for learning, after the test is finished, whether the model was suitable for buffet loads tests. It would be better, of course, to know this before making the test. As a guide for that purpose, the following procedure is suggested. With tabulated flutter coefficients, make a rough estimate of the aerodynamic damping. In order to approximate the structural damping, measure the damping of the wing in a still-air vibration test. If the measured structural damping is as low as 1/10 of the estimated

aerodynamic damping, the model will probably be satisfactory. Experience in transonic tunnels operating at atmospheric stagnation pressure has shown that solid metal wings are generally satisfactory if the wing-fuselage joints are tightly clamped. Difficulty has been experienced when insufficient joint fixity resulted in high structural damping and when tests were run at such low dynamic pressures that the aerodynamic damping was too low.

Model frequency requirements - Another requirement for the buffet model is that the vibratory mode shape and reduced resonant frequency be the same as those for the airplane. Of these two quantities, simulation of the resonant frequency probably is the more important. The power spectral density of the wing-root bending moment is plotted as a function of the reduced frequency in figure 30 for an airplane and for a 0.075-scale model of the airplane at the same Mach number. The vibration of the airplane wing is concentrated in the first symmetrical bending mode. This is also the case for the model; furthermore, the reduced resonant frequency for the model is about the same as that for the airplane. This comparison is interesting because the model has a standard solid-metal force-test wing. In the design of this model, there was no consideration whatever given to the resonant frequencies of the wing and yet frequency simulation of the first symmetrical bending mode was obtained. This seems to be a normal characteristic of solid-metal models of fighter-type airplanes (at least in the absence of external stores); thus, the simulation of wing resonant frequency for such airplanes presents no serious model design problems.

In the case of the tail, however, the situation is very different. The power spectral density of the bending moment at the root of a horizontal stabilizer is plotted as a function

of the reduced frequency in figure 31 for an airplane and for a 0.25-scale model of the airplane. Again, there was no attempt in the model design to simulate the frequency characteristics of the airplane. For the airplane, the most prominent mode is associated with fuselage torsion but this mode is hardly visible on the model, presumably because the model fuselage is more rigid than that of the airplane. Note also that, for the model, the frequencies of the various vibration modes are very different from those for the airplane. Buffet bending moments measured on this model stabilizer would probably bear little relation to those measured on the airplane. It appears then that, although force-test models are usually satisfactory for the study of wing buffet loads, the scaling of tail buffet loads will require models specially designed for that purpose.

Comparison of Flight and Wind-Tunnel Results

Wing-root bending moments during buffet have been measured in flight on a thin, unswept wing, research airplane (X-1E). They have also been measured on two models of different sizes in two different wind tunnels. When the fact that buffeting is inherently a random process is considered, the agreement between flight and wind-tunnel results has been satisfactory. Apparently, flight buffet loads can be estimated from wind-tunnel results, at least for simple wings. It follows from this result that wind tunnels can also be used to study the effect of airplane modifications on the buffet loads.

CONCLUDING REMARKS

In summary, I have attempted to illustrate briefly the role of aeronautical research in airplane design. In the future, we can see the solution of current problems, but more importantly we can predict even more. The role of aeronautical research in the design of the airplane, missile or space vehicle will require the best efforts of all.

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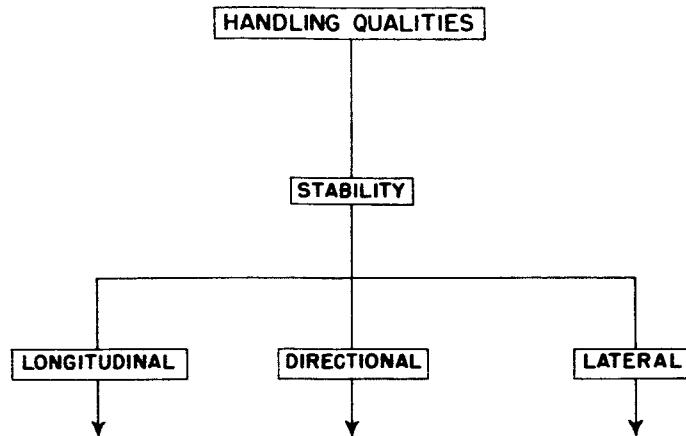


Figure 1

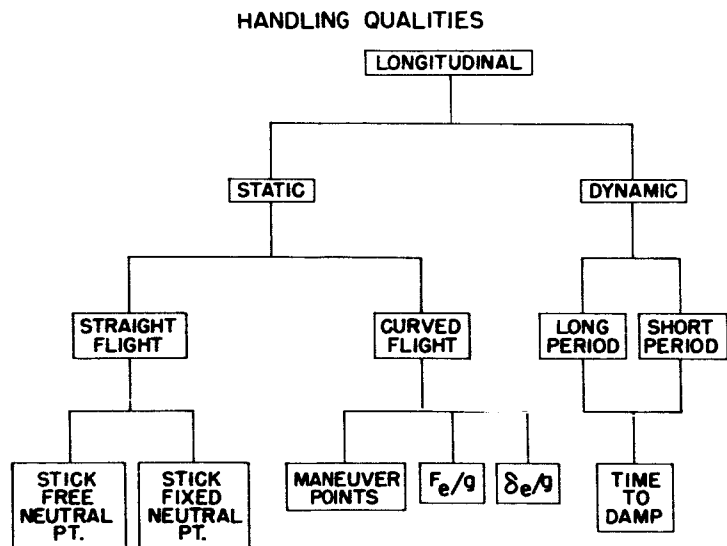


Figure 2

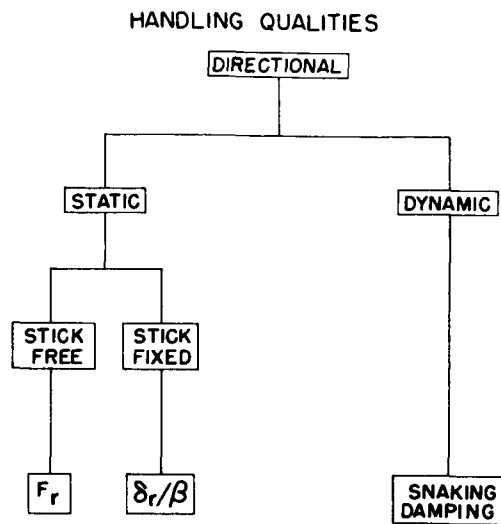


Figure 3

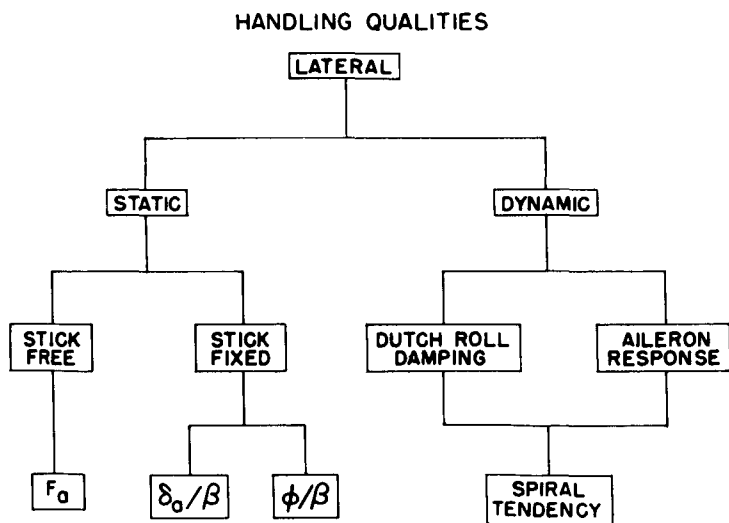


Figure 4

FLOWS ABOUT THIN WING SECTION

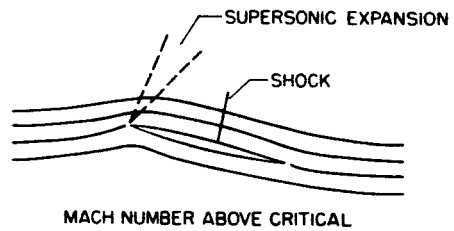
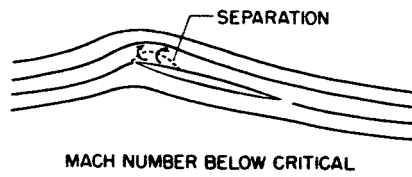


Figure 5

VORTEX FLOWS ABOUT FINITE WINGS

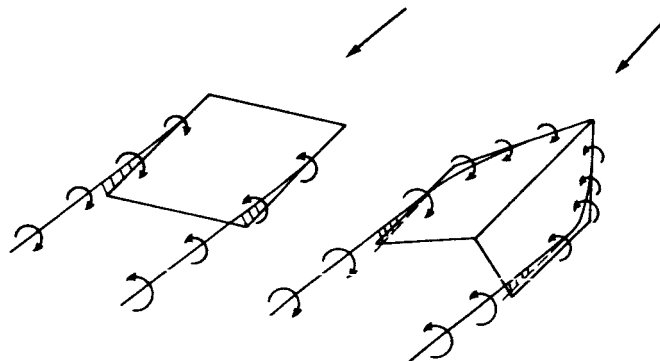


Figure 6

LINES OF MINIMUM PRESSURE ON A LIFTING BODY

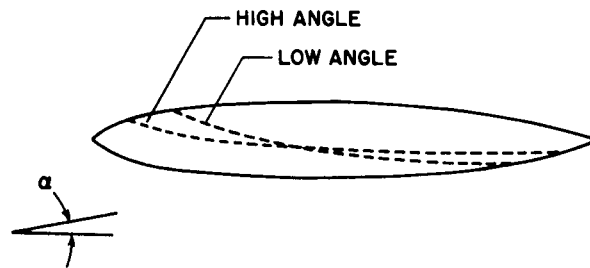


Figure 7

EFFECT OF VARIOUS BODY MODIFICATIONS

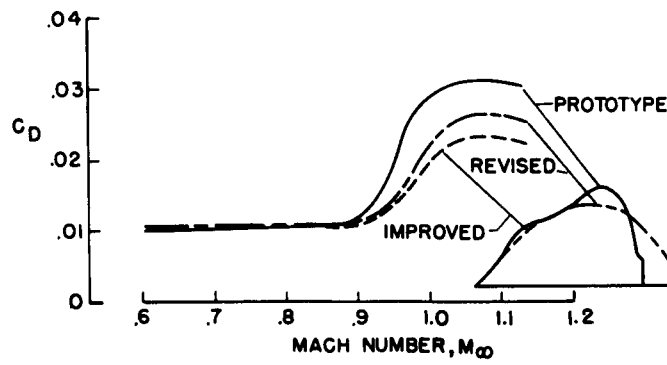


Figure 8.

NOZZLE PERFORMANCE

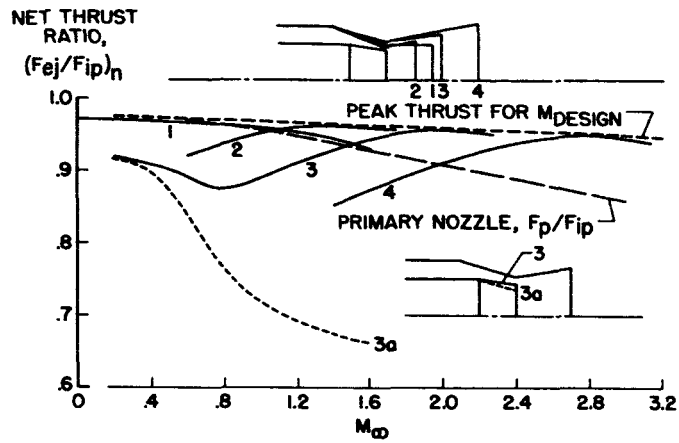


Figure 9

EFFECTS OF JET ON FUSELAGE DRAGS

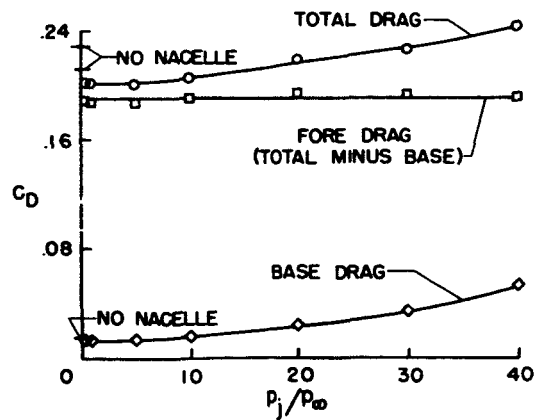
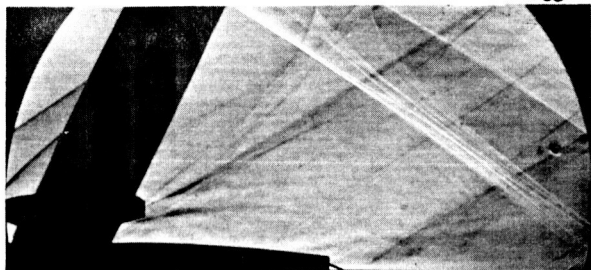


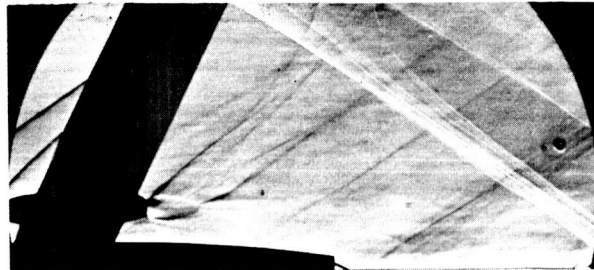
Figure 10

CONFIGURATION FLOW FIELD

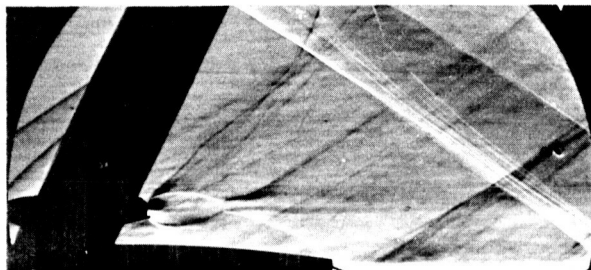
$$M_{\infty} = 1.94$$



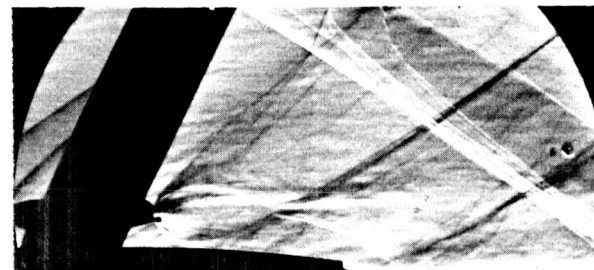
JET OFF



$$p_j/p_{\infty} = 10$$



$$p_j/p_{\infty} = 20$$



$$p_j/p_{\infty} = 40$$

Figure 11.

EFFECTS OF JET ON FUSELAGE DRAGS

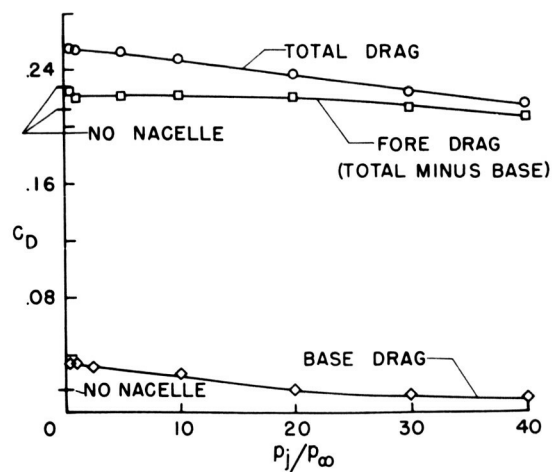
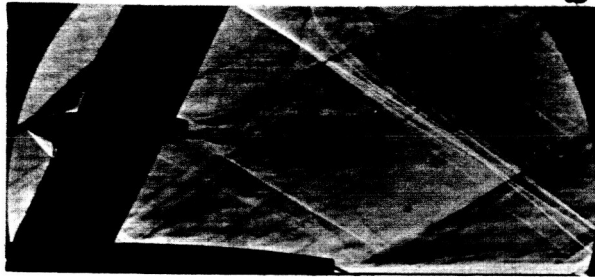


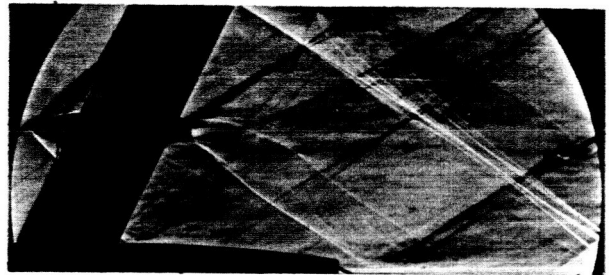
Figure 12

CONFIGURATION FLOW FIELD

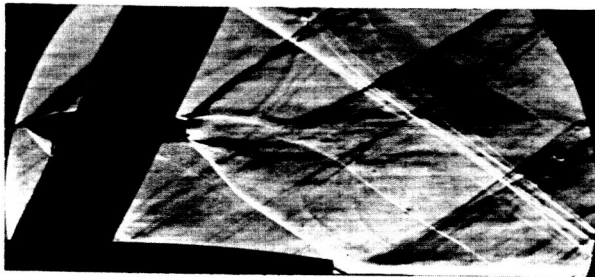
$M_\infty = 1.94$



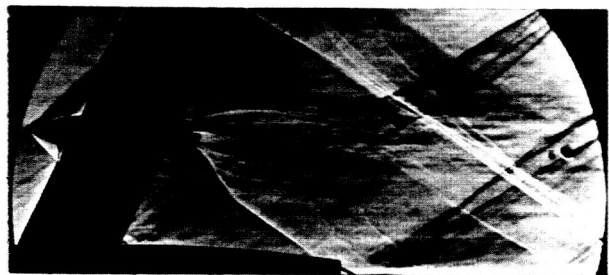
JET OFF



$p_j/p_\infty = 10$



$p_j/p_\infty = 20$



$p_j/p_\infty = 40$

Figure 13.

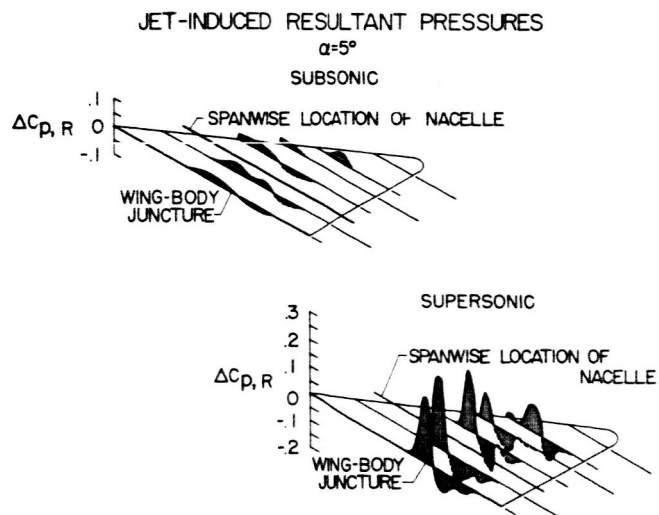


Figure 14

JET-INTERFERENCE EFFECT ON SPAN LOADING $\alpha=5^\circ$

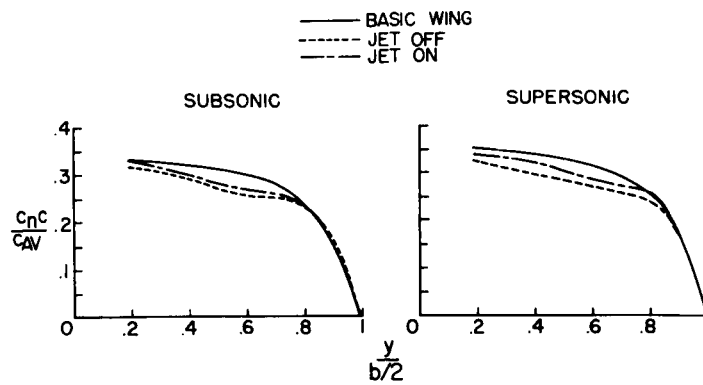


Figure 15

FLOW PHENOMENA ASSOCIATED WITH JET EXHAUST

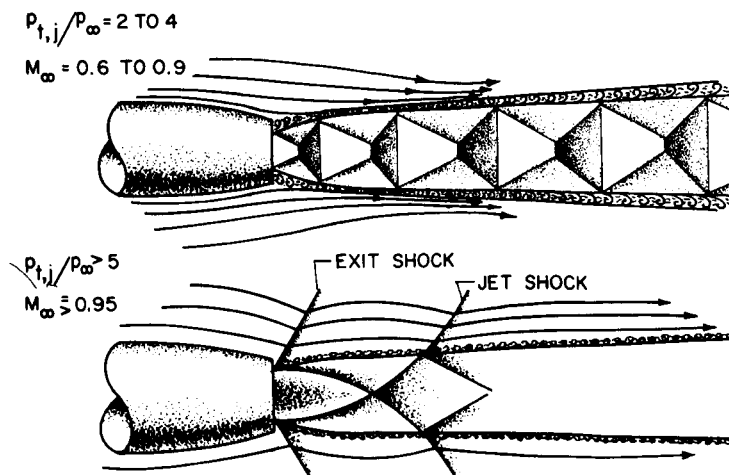


Figure 16

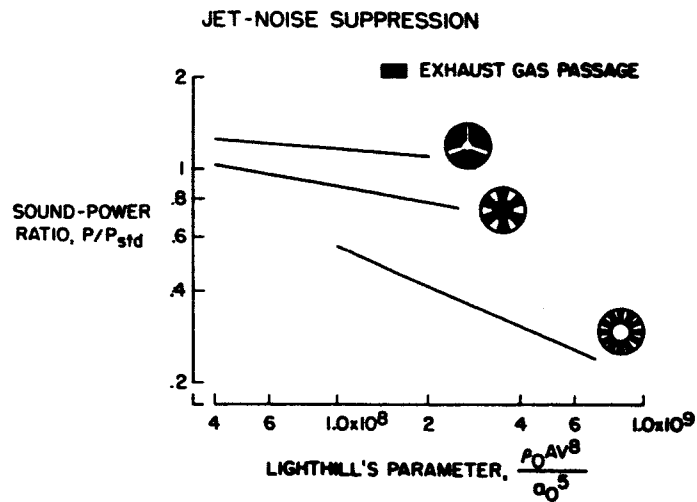


Figure 17.

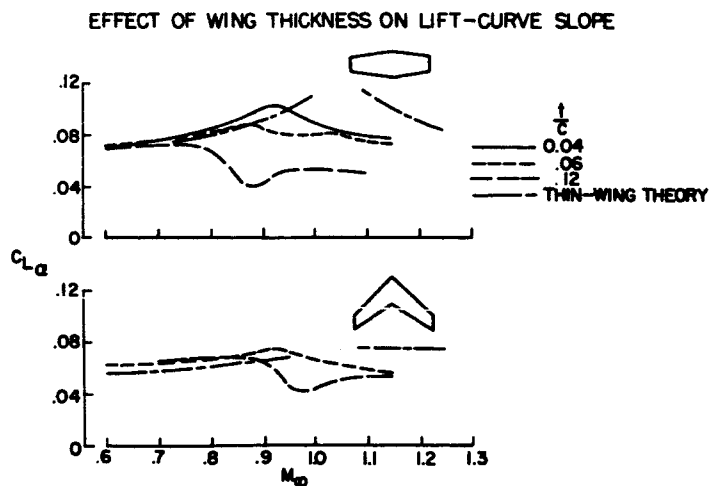


Figure 18

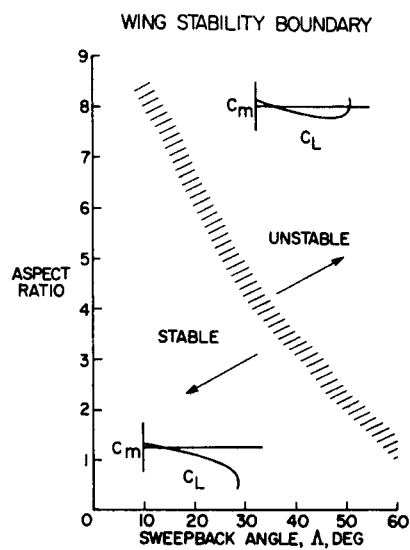


Figure 19

CONTRIBUTION OF DAMPING DERIVATIONS
IN ONE-DEGREE-OF-FREEDOM PITCHING

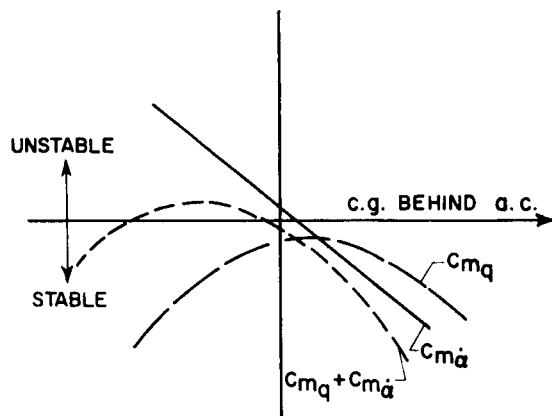


Figure 20.

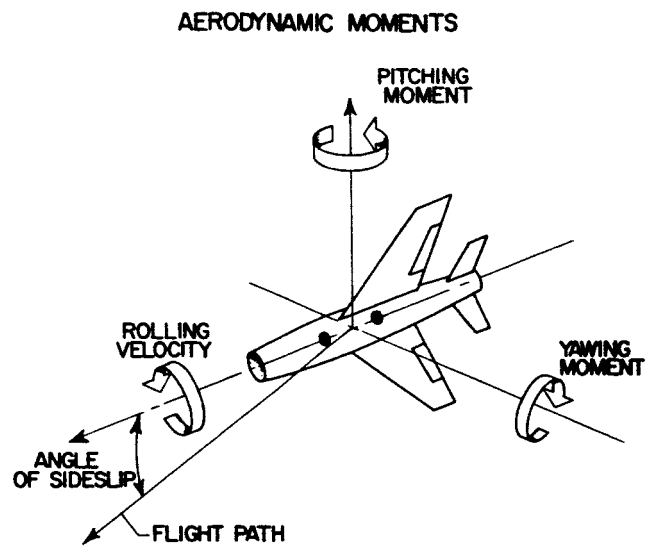


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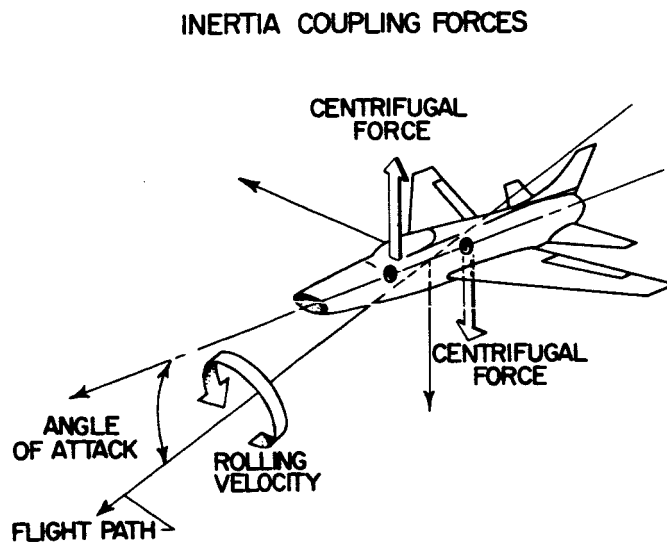


Figure 22

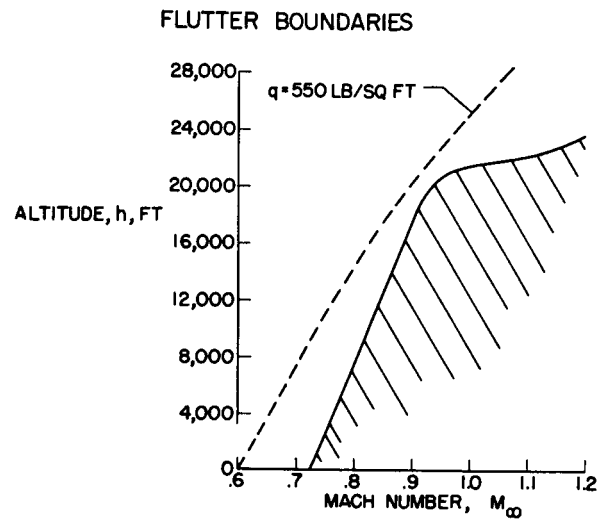


Figure 23

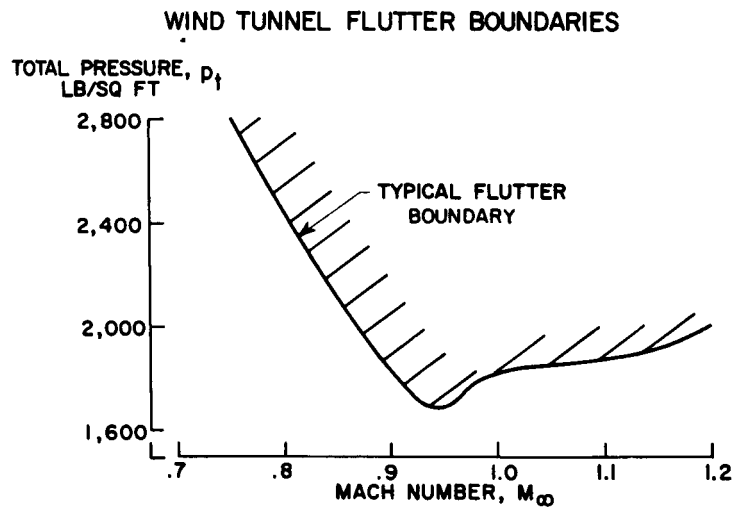


Figure 24

SCHEMATIC DRAWING OF LANGLEY TRANSONIC BLOWDOWN TUNNEL

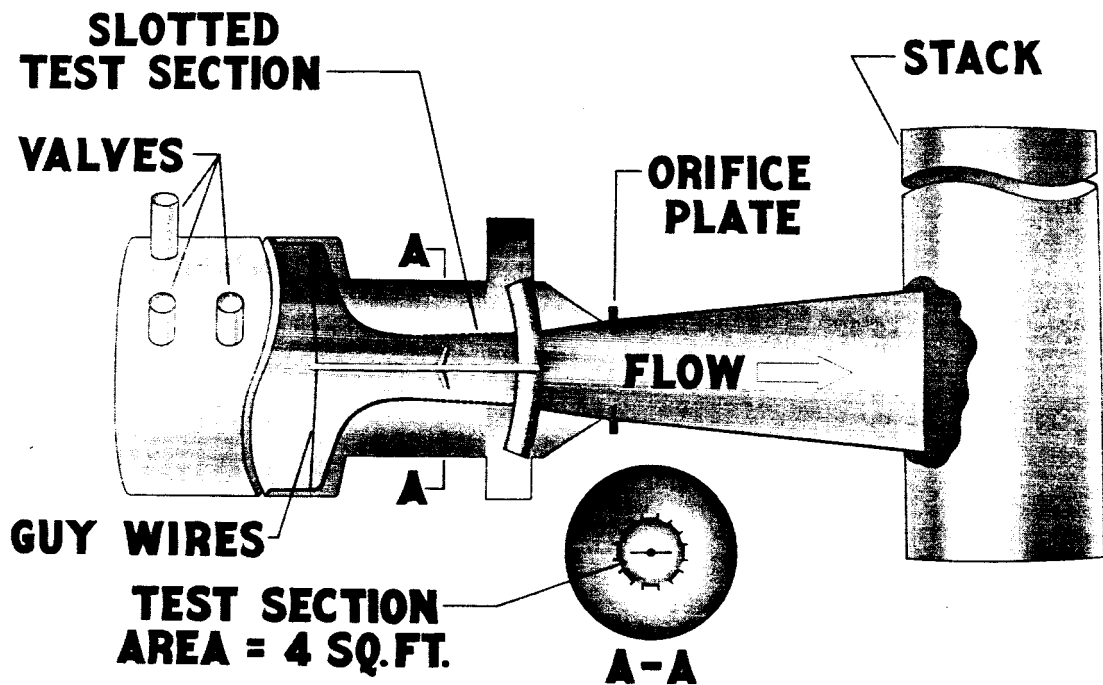


Figure 25.

TOWED-MODEL TEST TECHNIQUE

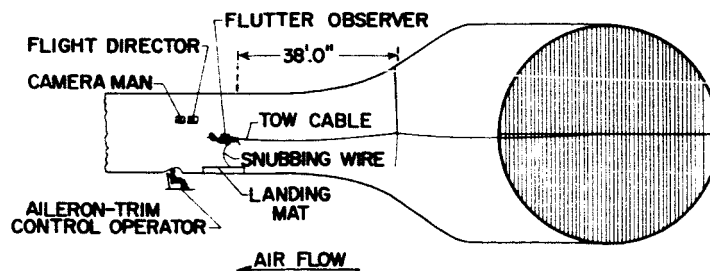


Figure 26.

TOWED-MODEL TEST TECHNIQUE

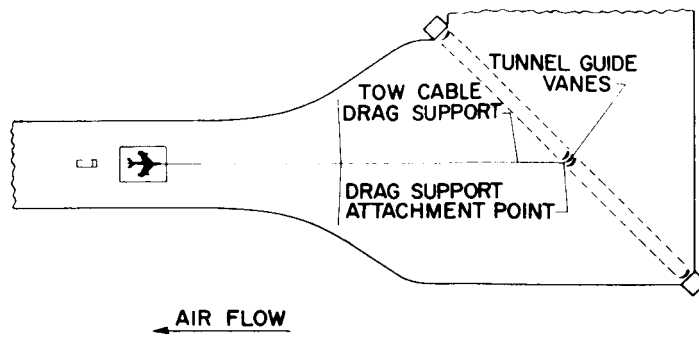
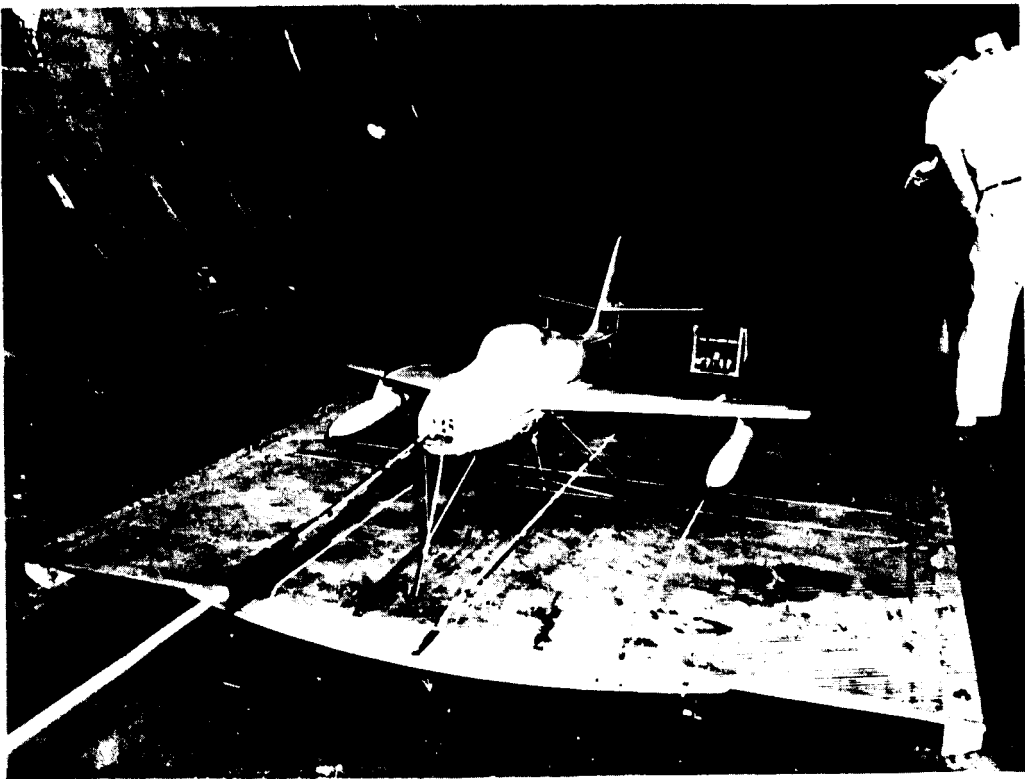


Figure 27

TOWED FLUTTER MODEL



THREE-QUARTER FRONT VIEW

Figure 28

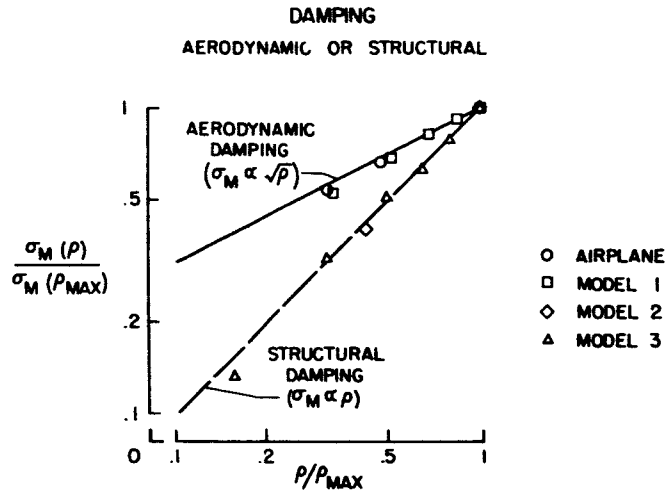


Figure 29

MODEL DESIGN
SIMULATION OF WING NATURAL FREQUENCY

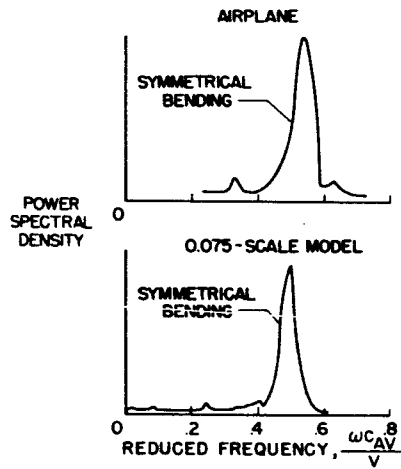


Figure 30

SUPPORT SYSTEM
EFFECT ON MODEL BUFFET MODES

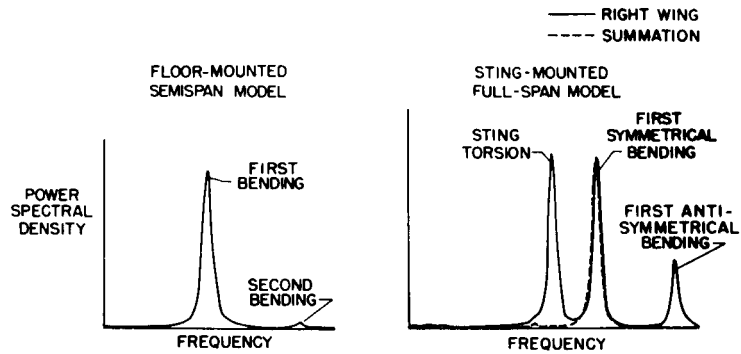


Figure 31.